RM-L9G06

[redacted]

Copy
RM L9G06

NACA RM L9G06

NACA

RESEARCH MEMORANDUM

PRELIMINARY ANALYSIS OF AXIAL-FLOW COMPRESSORS HAVING SUPERSONIC VELOCITY AT THE ENTRANCE OF THE STATOR

By Antonio Ferri

Langley Aeronautical Laboratory
Langley Air Force Base, Va.

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

WASHINGTON

September 12, 1949

[redacted]

 [redacted]

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

RESEARCH MEMORANDUM

PRELIMINARY ANALYSIS OF AXIAL-FLOW COMPRESSORS HAVING SUPERSONIC VELOCITY AT THE ENTRANCE OF THE STATOR

By Antonio Ferri

SUMMARY

A supersonic compressor design having supersonic velocity at the entrance of the stator is analyzed on the assumption of two-dimensional flow. The rotor and stator losses assumed in the analysis are based on the results of preliminary supersonic cascade tests. The results of the analysis show that compression ratios per stage of 6 to 10 can be obtained with adiabatic efficiency between 70 and 80 percent.

Consideration is also given in the analysis to the starting, stability, and range of efficient performance of this type of compressor. The desirability of employing variable-geometry stators and adjustable inlet guide vanes is indicated. Although either supersonic or subsonic axial component of velocity at the stator entrance can be used, the cascade test results suggest that higher pressure recovery can be obtained if the axial component is supersonic.

INTRODUCTION

A detailed discussion of a supersonic axial-flow compressor design in which deceleration of the flow from supersonic to subsonic velocity occurred in the rotor was presented by A. Kantrowitz in reference 1. This paper also included a brief discussion of a type of supersonic compressor in which the deceleration of the flow from supersonic to subsonic velocity occurred in the stator, and the possibility of obtaining high compression ratio per stage with this type of compressor was realized. However, this type was not investigated in detail because of lack of information pertaining to the problems of supersonic turning and diffusion and of interference effects. An analysis of the velocity diagram of such a compressor was presented in reference 2; however, the analysis was not extended to ranges of compression ratio considered in

[redacted]

the present analysis, and no attempt was made to solve any of the design problems.

Recent preliminary supersonic tests of two-dimensional diffusers and turning passages have indicated that the basic flow problems of compressor design utilizing supersonic flow at the stator entrance can be solved. Information bearing on the problem of efficient diffusion in the stator with inlet Mach numbers in the range of 2 to 3 is contained in reference 3. Three-dimensional diffusers are shown in reference 3 to be capable of a pressure-recovery ratio of 0.75 at $M = 2.50$ for aerodynamic designs in which the compression of the air entering the diffuser inlet does not interfere with the external flow. In these diffusers the starting limitations were eliminated by use of external compression during starting. Because external compression cannot be used in the stator of a compressor, a variable-geometry stator is necessary if pressure recovery values as high as those achieved in the three-dimensional inlet diffusers of reference 3 are to be obtained. Recent unpublished preliminary tests of a two-dimensional variable-geometry diffuser model representing a stator passage indicated a pressure-recovery ratio of 0.74 at $M = 2.51$, a value which is about equal to that obtained in the three-dimensional case. The problem of turning the air efficiently through large angles in the rotor is discussed in reference 4 which shows that a pressure-recovery ratio of about 0.95 is obtainable in a 90° turn at $M = 1.71$.

The purpose of this paper is to analyze the performance and to discuss the blade design problems of the supersonic stator type of compressor in the light of the information now available regarding the performance of supersonic turning passages and supersonic diffusers. The analysis and the experimental work both involve, for the most part, only two-dimensional flow and, therefore, are preliminary in character. No attempt has been made as yet to solve the three-dimensional flow problems. For example, the presence of large centrifugal forces could appreciably modify the results predicted by the present analysis.

SYMBOLS

θ	turning angle in rotor
β	turning angle in entrance vanes (fig. 1)
M_1	Mach number in front of rotor
M_2	rotational Mach number at entrance of rotor

M_2' rotational Mach number at exit of rotor

M_3 relative Mach number at entrance of rotor

M_4 relative Mach number at exit of rotor

M_5 absolute Mach number at exit of rotor and at entrance of stator

M_6 axial Mach number at entrance of stator

M_7 Mach number at exit of stator (less than 1)

P.R. compression ratio for stage

η adiabatic efficiency

ΔT increase in stagnation temperature across compressor

T_o stagnation temperature in front of entrance vanes

P_{o_n} stagnation pressure at Mach number M_n where n represents conditions from 1 to 7 indicated in figure 1

c_p specific heat at constant pressure

R gas constant

δ deviation of velocity direction at exit of rotor

ANALYSIS OF VELOCITY DIAGRAM

Consider a compressor design in which the tangential speed of the rotor is supersonic and the entrance velocity in the stator is also supersonic (fig. 1). Two different types of such compressors can be designed to correspond to two different types of velocity diagrams: a velocity diagram in which the velocity relative to the rotor is supersonic at the rotor exit $(M_4 > 1)$, and a velocity diagram in which this velocity is subsonic with respect to the rotor $(M_4 < 1)$ but still supersonic with respect to the stator $(M_5 > 1)$. Both types of compressors present good possibilities for practical applications and present similar problems of aerodynamic design.

These compressor types consist of the following three parts (fig. 1):

(1) Entrance vanes which change the direction of the flow through an angle β (positive when the air is turned against the direction of rotation of the rotor).

(2) Rotor blades which turn the flow through a large angle. The stream Mach number entering the rotor is M_1 and the rotational Mach number of the rotor M_2 is larger than 1.00; therefore, the entrance relative Mach number M_3 is larger than 1.00. The flow velocity in the rotor changes direction and magnitude, but remains supersonic in the first type considered; whereas, it becomes subsonic in the second type. The absolute velocity leaving the rotor is supersonic for both types.

(3) Stator blades which act as supersonic diffusers in which the air enters at the Mach number M_5 corresponding to the absolute exit Mach number from the rotor. The speed of the air is reduced to a subsonic value and then turned in the axial direction.

Important variables in this design are the entrance Mach number M_1, the angle β of the turning in the entrance vanes, the rotational speed of the rotor M_2, the turning angle in the rotor θ, and the variation of the Mach number in the rotor M_4/M_3.

In order to determine the effects of different parameters of the velocity diagram, some values of stagnation pressure losses must be assumed for the rotor and for the stator. Losses occur also in the entrance vanes but are assumed to be small and have been neglected in this preliminary analysis.

Compressor having supersonic relative velocity at exit of rotor - type 1.- The assumed values of the losses in stagnation pressure in the rotor are based on the preliminary tests of turning passages discussed in reference 4. Tests of a passage that turned the air 90° at an entrance Mach number M_3 of 1.71 indicated that a pressure recovery of 0.95 is possible for a ratio M_4/M_3 near 1. The pressure recovery decreased for lower values of M_4/M_3. The test Reynolds number (about 9.16×10^6 with respect to the chord) was appreciably higher than might be expected in the usual supersonic compressor. The ratio of blade span to chord (2:5) was relatively low.

No test data are available for turning angles other than 90° and entrance Mach numbers other than 1.71. The losses in the passage have been assumed in the analysis to be independent of the turning angle and of the entrance Mach number M_3. Actually, the losses probably increase somewhat with the increase of turning angle. Also, the possible presence of detached shocks and of local subsonic zones can change the value of the losses when the value of M_3 becomes smaller than 1.71. The value of M_3 in the analysis is of the order of 1.71 for $M_2 = 1.25$ and M_1 of the order of 0.85. The value of M_3 changes when the entrance Mach number and the rotational velocity change (for $M_1 = 0.40$ and $M_2 = 1.25$, $M_3 = 1.41$, and for $M_1 = 1.20$ and $M_2 = 1.30$, $M_3 = 1.98$). The values assumed for the pressure recovery P_{o_4}/P_{o_3} in the rotor are shown in figure 2. The values assumed for the pressure-recovery ratio in the rotor (shown in fig. 2) are somewhat lower than the highest values given in reference 4. The data of reference 4 are considered somewhat pessimistic because of the low values of aspect ratio; however, other losses not considered in the analysis can be expected in the actual rotating machine.

The two-dimensional experimental results for the stator show that values of pressure recovery P_{o_7}/P_{o_5} of the order of 0.88 for $M_5 = 2.01$ and 0.74 for $M_5 = 2.51$ are obtainable. Values of pressure recovery of 0.87 and 0.71 have been assumed at $M_5 = 2.00$ and $M_5 = 2.50$ for the stator, and a linear variation of pressure recovery with Mach number has been assumed in the analysis (fig. 3).

With these assumptions, compression ratios and adiabatic efficiency were determined for systematic variations in the velocity diagram. With the compression ratio P.R. defined as the ratio between stagnation pressure at the exit of the stator and stagnation pressure at the entrance of the compressor, the adiabatic efficiency η was determined from the equation

$$\eta = \left(\text{P.R.}^{R/c_p} - 1\right)\frac{T_o}{\Delta T} = \left(\frac{\gamma T_o}{\Delta T}\right) \qquad (1)$$

where ΔT is the increase in stagnation temperature across the compressor.

Some results of the analysis are shown in figure 4, in which the compression ratio and the adiabatic efficiency are presented as functions of the rotor turning angle for different turnings in the guide vanes. The rotational Mach number chosen for the compressor is $M_2 = 1.25$ and the entrance Mach number is $M_1 = 0.85$. For this Mach number, the entering mass flow is only 2 percent less than the maximum, but it is believed possible to design nonchoking guide vanes for this condition. A small compression is assumed in the rotor corresponding to $\frac{M_4}{M_3} = 0.90$. The compression ratio increases when the turning angle in the rotor increases, and the increase is of the same order as the increase of the turning angle. The compression ratio increases also if guide-vane turning is used. For the pressure recovery assumed for the rotor and the stator, the adiabatic efficiency decreases somewhat when the compression ratio increases. The decrease in adiabatic efficiency due to the turning in the guide vanes (fig. 4) probably is smaller than should be expected, especially for large angles of turning in the guide vanes because in the calculations no losses are considered in the guide vanes.

In figure 5, the compression ratio and the adiabatic efficiency are shown as functions of the entrance Mach number for different values of rotational speed. The turning angle in the guide vanes is assumed equal to 15° and in the rotor, 100°. The assumed compression in the rotor corresponds to $\frac{M_4}{M_3} = 0.85$. The maximum compression ratio is obtained in the range of M_1 between 0.8 and 1.0, and the maximum efficiency is obtained in the range of low entrance velocity. The compression ratio increases with increasing rotational speed.

In figure 6, the compression ratio and adiabatic efficiency as functions of the turning in the guide vanes for different values of compression ratio in the rotor are given. The entrance Mach number is 0.85 and the rotational Mach number, 1.25, and the turning angle, 100°. The compression ratio increases when the turning in the guide vanes increases. Some compression in the rotor $\left(\frac{M_4}{M_3} < 1\right)$ is desirable. The optimum amount of compression depends on the magnitude of the losses in the rotor and in the stator and for the assumed values corresponds to a value of $\frac{M_4}{M_3} \approx 0.90$. The efficiency decreases slightly when the turning in the guide vanes increases. The decrease probably is larger than shown in the figure, because the losses in the guide vanes have been neglected.

This analysis is preliminary in character but indicates that with a correct choice of the different parameters, adiabatic efficiency of the order of 70 to 80 percent and compression ratio of the order of 6 to 10 can probably be expected in this aerodynamic design.

Compressor having subsonic relative velocity at exit of rotor - type 2.- This compressor is geometrically similar to the design previously discussed (type 1). Guide vanes can be used in front of the rotor. The stream entering the rotor has supersonic velocity relative to the rotor (fig. 1). The rotor passage changes the direction of the entering stream through a large angle (of the order of 90°). The passage is a converging-diverging channel similar to the passage considered for type 1, but in this case, transition from supersonic to subsonic velocity occurs in the rotor and, therefore, the exit relative velocity M_4 is subsonic.

The absolute velocity M_5 is supersonic and the stator is a supersonic diffuser similar to the diffuser considered for the preceding case. However, the value of the entering Mach number is lower than the value considered in type 1; therefore, the pressure recovery that can be expected in the stator is larger than for the type-1 design. The tangential component of the absolute velocity at the exit is smaller and, therefore, the centrifugal forces are reduced with respect to the centrifugal forces existing in the type-1 case.

A few unpublished preliminary tests of the turning passage model 3 of reference 4 (fig. 7) with subsonic exit velocities have been made. Subsonic velocity at the exit was obtained by increasing the back pressure, and pressure-recovery ratio of the order of 0.84 was measured for an entrance Mach number of 1.71.

A pressure-recovery ratio of the order of 0.90 is considered possible for the stator in view of the relatively low stator entrance Mach numbers $M_5 < 2.00$ for the type-2 compressor.

Some values of possible compression ratio and efficiency have been calculated by assuming 0.80 and 0.90 pressure recovery in the rotor and 0.88 in the stator. The exit relative velocity in the rotor has been arbitrarily assumed equal to 0.37 of the value of the entrance relative velocity. The results are shown in figure 8. The results of these calculations (fig. 8) indicate that values of the compression ratio of the order of 7 and efficiency of the order of 0.80 to 0.85 can be expected for a turning angle of 100° and pressure recovery in the rotor of 0.80. The entrance velocity relative to the rotor for $M_1 = 0.85$ is of the order of 1.70, and the entrance Mach number in front of the stator for $M_2 = 1.25$ is of the order of 1.50.

ROTOR BLADE DESIGN

The previous analysis indicates that large turning angles of the order of 90° are required in the rotor and that some deceleration of the flow in the rotor is desirable.

A rotor blade for the case in which the exit velocity relative to the rotor is supersonic can be designed as follows. Consider first a two-dimensional passage in which the relative entrance velocity is supersonic. By means of the characteristics system, a turning passage that permits the flow to turn without formation of strong shocks and without large losses can be designed (reference 4). In the design of the wave pattern, a gradual decrease in speed can be obtained along the channel; however, the dimensions of the channel in the direction of the rotational velocity at the entrance and at the exit of the rotor must be the same in order to obtain a blade with sharp leading and trailing edges (fig. 8). When the channel is assumed to be two-dimensional, a given value of the ratio M_4/M_3 corresponds to a given velocity diagram and to a given value of the pressure recovery.

In order to vary the value of the ratio M_4/M_3 with a given velocity diagram, a contraction of the channel in the radial direction must be used. Contracting the channel radially permits the desired value of the ratio M_4/M_3 to be chosen without altering the velocity diagram.

The rotor blade cannot have zero thickness in the region of the leading edge and, therefore, a shock wave of some intensity must be accepted at the leading edge. The necessity of a wedge of finite dimension at the leading edge requires also that the relative entrance velocity be high enough to give an attached shock, in order to avoid introducing additive losses associated with the detached shock. The presence of the detached shock would reduce the efficiency somewhat but would not radically change the general performance of the compressor. In the tests discussed in detail in reference 4, a 10° wedge was used for the blade, and the ratio of the average exit Mach number M_4 with respect to the entering Mach number changed in the range between 0.85 and 1.05 when the shape of the passage changed. A pressure recovery of 0.85 to 0.95 was obtained for $M_3 = 1.71$ and a turning angle of the order of 90°. The blade considered in reference 4 has a thickness ratio of about 0.12 and a shape which is practical for the structural requirements (fig. 7).

The velocity leaving the rotor of the type-1 design is supersonic, but the component in the direction of the axis of the rotor can be supersonic or subsonic depending on the velocity diagram. The axial component of the exit velocity is supersonic for the lower range of turning angles in the rotor and becomes subsonic for the higher range of turning angles. For $M_1 = 0.85$ and $M_2 = 1.25$, the axial component becomes supersonic for a turning angle of about 80°. If M_2 increases, the axial component tends to become supersonic also for high values of turning angles.

A similar design can be used for the rotor passage of the compressor of the second type. As previously mentioned, reasonable pressure recovery (0.84) was measured in preliminary tests of a possible rotor passage when the exit velocity was subsonic $(M_4 < 1.00)$. However, the zone of separation near the convex surface in this case was larger than for the case of supersonic flow at the exit of the passage ($M_4 > 1.00$). The presence of a large wake at the exit of the rotor would probably produce other losses in the stator and could possibly have important effects in an actual machine in which three-dimensional phenomena exist.

STABILITY OF FLOW CONDITIONS IN A ROTOR WITHOUT CONTRACTION

The previous analysis shows that large pressure ratios per stage and practical values of adiabatic efficiency can be obtained for this compressor when operating at design conditions. However, the compressor, in order to be practical and stable, must operate for flow conditions on either side of the design point. The effects of variation of the rotational speed and of the entering Mach number must therefore be considered. In addition, it must be shown that the machine is capable of starting and establishing the design conditions.

For the analysis of the phenomena related to stability and to starting of the rotor, the velocity diagram considered is one for which the axial component of the entering Mach number M_1 is subsonic (fig. 1). The turning passage corresponds to the design of figure 7 and consists of two walls, one of which is parallel in the region of the leading edge to the design stream direction. First, the turning passage is assumed to be a channel in which no contraction exists and the minimum cross section is at the entrance. If the entrance Mach number M_1 is reduced and the rotational velocity is not changed, the relative velocity decreases and changes direction (M_3'; fig. 9(a)). As shown in reference 1, an expansion produced at B (fig. 9(a)) travels upstream because the axial component of the velocity is subsonic. The expansion produced at B changes the direction and velocity of the stream as it moves outside of the passage. A family of expansion waves from the rotor blades thus changes the direction and magnitude of the stream from OL_1 to OL, the design condition for which no waves occur. The reduction of axial speed is thus neutralized by the expansion waves that are produced by the rotor; the flow tends to return to the conditions fixed by the rotor design. In a similar way, if the entrance Mach number M_1 is increased, a compression wave is produced at B (fig. 9(b)), and the flow tends to return to the conditions fixed by the rotor design. This stable characteristic depends on the expansion or compression waves from point B.

When the axial velocity is decreased, the shock at the lip of the surface of the blade inclined to the stream increases in strength and for low values of axial velocity may become detached. For this condition, further analysis is necessary to determine whether it is still possible for expansion waves from the lip to travel ahead of the rotor, thereby stabilizing the flow. The problem may be approached by considering schematically a wedge in a supersonic stream at a large angle of attack (fig. 10). The detached shock produces subsonic flow behind the shock. At the upper surface BC, behind the leading edge B, the flow produces a vortex and local separation. The flow in the upper zone, however, expands and the velocity outside of the zone of separation increases again to a supersonic value. If the separation is localized near the leading edge, the Mach number along the surface BC behind the region of separation differs only slightly from the Mach number that would exist if no detached shock were present and the flow had expanded from the direction of the undisturbed stream to the direction BC. (See references 4 and 5.) In this case, expansion waves are produced in the region of the leading edge which neutralize the shock and then extend to infinity and turn the undisturbed velocity to the direction BC. When the angle between the direction of the undisturbed velocity and the surface BC is very large, the separation of the flow along BC is no longer localized near the leading edge but extends downstream. In this case, the flow does not assume the direction BC, and expansion waves transmitted upstream are weaker or may be nonexistent. The possibility of large separation increases when an adverse pressure gradient exists along the surface BC. However, as long as the detached shock is not accompanied by extensive separation over a large part of the rotor blade, expansion waves will be produced and the mechanism by which stable operation occurs will be the same as that described for the attached shock conditions considered in the preceding paragraph.

When the separation is not localized but occurs along all the upper surface BC of the blade, the intensity of the expansion waves tends to decrease; and when no expansion waves are transmitted upstream, the flow tends to remain of subsonic type inside the passage. The tests of references 4 and 5 show that expansion waves are produced even for angles of attack as high as 40°. Thus, the mechanism required for stability will exist even when the entering Mach number M_1 has very low values.

Any reduction in rotational speed of the rotor produces a change in direction of the entrance velocity. Compression waves are produced at the lip of the blade which travel upstream and reduce the velocity. (See reference 1.) For steady conditions, the relative velocity decreases and becomes OL_1 (fig. 9(b)). The compression waves, therefore, tend to reduce both the relative entrance velocity and the axial velocity.

If the reduction of rotational speed is small, and the entrance Mach number OL_1 is large enough, the shock produced at B by the surface BA is still attached; for a large reduction of rotational speed, however, a detached shock occurs at B. The steady conditions correspond, as with the attached shock, to OL_1 because for this condition only can the expansion waves produced at B neutralize the detached shock; that is, this is the only condition for which no waves are transmitted upstream.

In order to allow a reduction of rotational velocity without a decrease in efficiency due to shock detachment, the wedge at B should be smaller than the angle of maximum deviation corresponding to the design condition. For example, if the wedge angle is 10°, the minimum Mach number that permits an attached shock is 1.43. If, then, the relative Mach number is 1.70 and the rotational speed corresponds to Mach number 1.25, the rotational velocity can be reduced 15 percent before a steady detached shock is produced in front of the blade.

PROBLEM OF STARTING SUPERSONIC FLOW IN A ROTOR HAVING A CONVERGING-DIVERGING PASSAGE

The rotor can be designed with passages having the minimum cross section at the entrance, in which case no particular problem exists in regard to starting the supersonic flow. Reference 4 shows, however, that the use of a converging-diverging passage in the rotor is preferable. When a contraction exists in the passage, starting problems similar to those of supersonic diffusers exist for the rotor and, therefore, the mechanism of establishing supersonic flow in the passage must be analyzed.

Consider first that the rotational velocity of the rotor is being gradually increased. For low subsonic speeds, the increase in rotational velocity causes an increase in the velocity within the rotor passages. This increase continues as the rotational velocity advances until sonic velocity is reached at the minimum section of the rotor passage. For higher rotational speeds, the direction of the inlet velocity relative to the rotor cannot be parallel to the upper surface of the rotor blade until the flow at the entrance of the rotor becomes supersonic because for this condition the continuity requirement cannot be satisfied. The axial component in this case must be smaller than for the case in which the stream flows parallel to the rotor blade and, therefore, the stream direction tends to produce expansions at the leading edge of the blades (reference 1). The expansions at the leading edge of the blade tend to increase the stream velocity. If the velocity could be increased by this means until the flow became parallel to the entrance of the blade, the Mach number after expansion would be larger than the Mach number M_3 which corresponds to the same rotational velocity of the rotor and to an axial velocity for which the entering stream is parallel to the upper surface of the rotor blade.

In this condition, the expansion waves tend to increase the Mach number of the stream entering the rotor and also the cross section of the stream tube entering the rotor. If the contraction of the rotor is too large to allow passage of the increased flow, compression waves would move upstream from the rotor throat and cancel the expansion waves produced at the leading edge of the blade. In this case, then, the entrance velocity relative to the rotor must actually remain inclined with respect to the rotor blade, and starting could not occur. If the speed is further increased to a sufficiently high supersonic value so that the continuity law is satisfied for the rotor contraction ratio, no compression waves could be transmitted upstream, and the expansion waves from the leading edge of the blades could then increase the velocity of the incoming flow until the relative velocity is parallel to the upper surface of the blade, and the starting of the supersonic flow would thus occur.

The mechanism of starting as previously described is complicated by the presence of local separation of the flow at the leading edge and by detached shock. The actual starting rotational Mach number for a given contraction ratio and passage design is affected by the losses due to the compression waves from the throat and from the losses due to friction and separation. In general, the losses due to compression waves and shock waves behind the expansion of the blade from the leading edge are less than the losses that correspond to a normal shock in front of the passage, but the differences can be small, especially if the wedge of the blade at the leading edge produces a detached shock and local separation at the leading edge.

The foregoing discussion indicates that starting of the supersonic flow in the rotor would occur for a contraction ratio larger than the theoretical limiting contraction ratio given by one-dimensional theory for an inlet diffuser for which a normal shock occurs ahead of the inlet prior to starting. The results discussed in reference 3 and preliminary results from variable-geometry stator-blade tests however, show that, although the starting limitation can be avoided by use of variable geometry, supersonic flow cannot be maintained for throat Mach numbers close to 1.00. Disturbances from the downstream flow and from the boundary layer have a destabilizing effect which causes the entrance flow to become subsonic if an attempt is made to operate with throat Mach numbers near unity. Experiments at $M = 1.65$ for a particular variable-geometry model in which appreciable disturbances were known to exist showed that the maximum contraction ratio for stable flow was less than the theoretical contraction ratio for the fixed-geometry case. These experiments also showed that the advantages of variable geometry decrease rapidly with decreasing Mach number. Thus it can be concluded that, for the relatively low supersonic Mach numbers considered for the rotor, any appreciable gain in contraction ratio over the theoretical values for the one-dimensional fixed-geometry inlet diffuser is improbable.

For Mach numbers less than the starting value, losses exist because of the waves in front of the rotor. For these conditions, adjustable guide vanes in front of the rotor are desirable. If the rotational velocity is too low for starting, the axial component of the entering velocity is small and, therefore, an appreciable amount of turning can be obtained without shock losses, especially if the guide vanes are in a zone ahead of the rotor in which the axial velocity is lower than that immediately in front of the rotor. The guide vanes can introduce a favorable rotation in the stream and thus decrease or eliminate the losses due to the waves in front of the rotor. Adjustable guide vanes also offer the possibility of eliminating the geometrical starting limitations in cases where these limitations actually control the starting process rather than the limitations introduced by the instability of the supersonic flow inside the passage.

REGULATION OF MASS FLOW

In supersonic compressors, when the axial component of the entering Mach number M_1 is subsonic and the flow inside the rotor is supersonic $(M_3 > 1)$, the magnitude of the entering velocity is determined by the rotational speed. No regulation of the mass flow entering the compressor is possible by means of throttling the flow downstream of the compressor. For the supersonic compressor with subsonic axial component of the entrance Mach number M_1, the regulation of the mass flow can be accomplished by changing the direction of the entering velocity by means of adjustable guide vanes. Turning the guide vanes allows the value of the turning angle β of figures 9(a) and 9(b) to be changed; for a constant rotational Mach number M_2, the direction of the entering stream M_3 can thus be changed. Expansion waves or compression waves are then produced at the leading edge of the compressor blades and correspond to an increase or to a decrease of the value of β. The waves change the amplitude of the entering Mach number M_1 and thereby change the volume of flow entering the rotor.

If the axial component of the entering Mach number M_1 is supersonic, the waves produced at the leading edge of the rotor blades are contained inside the passage. In this case the mass flow is established by the geometry of the passage ahead of the compressor and remains unaffected by any variations in the guide-vane settings. Regulation of the volume flow, however, can be obtained by adjustment of the guide vanes, which are now of the supersonic type, because the contraction ratio in the guide vanes changes with blade setting. If it is desired to vary mass flow where M_1 is supersonic, this result can be accomplished by means of a variable-geometry inlet duct ahead of the compressor similar to the one discussed in reference 3.

DESIGN OF STATOR AND INTERFERENCE EFFECTS

BETWEEN ROTOR AND STATOR

The tests of the inlets of reference 3 indicated that pressure recovery higher than the pressure recovery that can be expected in a convergent-divergent diffuser can be obtained for stream Mach numbers between 2 and 3 if the starting limitations are avoided by use of variable geometry. In some of the inlets of reference 3, compression occurs in front of the entrance and the external flow does not interfere with the internal flow; therefore, conditions similar to those required in a diffuser for supersonic compressors have been obtained. The steady conditions in these diffusers are similar to the conditions of the flow in a convergent-divergent diffuser; however, the mechanism of starting is different. The stream-tube contraction ratio used in the inlets of reference 3 is larger than the stream-tube contraction ratio used in a convergent-divergent diffuser. Although the starting mechanism used in a conical inlet cannot be used directly in a diffuser of the stator of a supersonic compressor, values of pressure recovery similar to those obtained for the conical inlets can be expected in a diffuser with all internal compression, if the contraction ratio is higher than that determined by the fixed-geometry starting conditions.

Consider a two-dimensional inlet such as that shown in figure 11. At first, the flow component normal to the entrance AB of the diffuser is assumed to be supersonic, in which case, the diffuser can be designed as follows: The upper surface AD of the lower blade is inclined at an angle to the stream Mach number M_5 and produces a shock wave. As an alternate to the shock wave from A, a family of compression waves can be substituted by using a surface AD that starts at A parallel to the incoming flow and curves gradually. For the design conditions, the shock wave or the Mach wave from A hits the lower surface of the other blade BE behind the leading edge B and is reflected. The contraction ratio of the passage can be decreased by turning the front part of each blade that constitutes the stator as shown in figure 11. The shock from A becomes weaker, and the reflected shock from the surface BE becomes weaker. For the position A' of figure 11, the surface AD produces an expansion, and the internal contraction is very small.

If, for the design conditions, the blades are turned in a direction opposite to the direction shown in figure 11, the shock wave or the compression waves from the lower blade move in front of the upper blade and, therefore, a family of compression waves tends to move upstream in front of the stator. This family of compression waves tends to decrease the entrance velocity and, therefore, tends to change the volume of the flow entering the stator. In this case, interference occurs between stator and rotor. In order to avoid such interference, the shock A and all the compression waves produced along AD must lie behind the line AB; therefore, the component of the stream Mach number M_5 normal to AB must be supersonic (shock at A). If this component is subsonic, the interference disappears only when the surface AD at the leading edge A is parallel to the direction of the incoming flow and the compression waves along AD originate far enough behind the leading edge so that the waves are confined inside the passage.

When compression waves or expansion waves move in front of the stator and reach the exit of the rotor they change the magnitude and direction of the stream leaving the rotor and, therefore, the velocity diagram is changed. For some ranges of intensity of the waves moving upstream from the stator, the equilibrium condition is obtained at the exit of the rotor, and the velocity distribution inside the rotor does not change. In this range, the waves moving upstream from the stator become steady waves attached to the exit of the rotor.

Consider a compression wave moving upstream from the stator (fig. 12(a)). The compression tends to reduce the velocity component in a normal direction to the wave and, in fixed coordinates, tends to change the velocity from the vector OB to the vector OB'. Because the rotational velocity is constant, the velocity in rotor coordinates must change from OC to OC'. When the compression wave reaches the exit of each passage it tends to move inside (wave EF) and is reflected at one of the surfaces of the blades and, therefore, another compression wave ED is produced that changes the direction of the velocity OC' to the direction OC and increases the pressure in the zone behind the wave. At the trailing edge F of the blade, equilibrium of pressure and direction must exist; therefore, an expansion wave is also produced at the trailing edge of each blade which neutralizes the reflected compression wave ED. This wave pattern cannot move upstream inside the passage because, in order to move upstream, the wave EF must be strong enough to develop into a normal shock corresponding to the velocity OC. Furthermore, no waves are reflected downstream. This wave pattern, therefore, becomes steady and the steady condition corresponds to the velocity OC'.

A similar phenomenon occurs if expansion waves are considered (fig. 12(b)). When either expansion waves or compression waves are produced by the stator and move upstream, the flow conditions at the end of the rotor change and steady waves are produced at the end of the rotor that stabilize the flow.

The equilibrium conditions between rotor and stator and the maximum intensity of the compression waves that can be trapped at the end of the rotor can be determined for each configuration from the preceding considerations. For every value of the vector OC (fig. 12(a)) representing the exit Mach number in rotor coordinates for the condition of zero interference, the line C'CC'' can be determined by considering that incoming waves have different intensity and by determining the intensity of the reflected waves. From the diagram C'CC'', the diagram B'BB'' in stationary coordinates can be determined. For the case of expansion waves moving upstream, the condition of equilibrium corresponds to a vector OB parallel to the entrance region of the stator PQ (fig. 12(a)). The same consideration is valid for compression waves.

When the component of the entering velocity normal to the entrance of the stator PR is supersonic, another condition of equilibrium can exist that corresponds to a shock wave at P along PR or inside the passage. This second condition is possible only for the velocity vector OB because if shock waves travel outside of the stator passage they tend to reduce the component in normal direction to the wave to subsonic values. When the axial component of the velocity OB is supersonic, a shock at P can be utilized in the diffuser to increase the pressure recovery. However, if the shock is along the line PR at the entrance of the passage, the condition of equilibrium is not stable because if the shock is disturbed from its original position and moves outside of the passage, the velocity normal to the line PR becomes subsonic and the original condition is not established again. In order to obtain stability, the shock should be slightly inside the passage. Because the entering Mach number is usually large, the angle between the shock and the entrance of the passage required is small. Assume, for example, that the Mach number entering the stator is 2.30 and a shock corresponding to a 10° deviation of the flow is produced at the tip of the blade. The angle of the shock is $34^{\circ}20'$. If an angle of 2° is chosen between the front of the shock and the line PR in figure 12(a) the shock is stable for a variation of tangential velocity of 10 percent. If the rotational component corresponds to $M = 1.25$, the shock is stable for a variation from $M = 1.25$ to 1.10. The shock will remain in the stable position for a larger decrease in tangential velocity if a corresponding decrease in normal velocity component also occurs. However, in order to utilize a diffuser having a shock at the leading edge of the blade, a variable-geometry stator is necessary to start the phenomenon. A variable-geometry stator is also required in order to have the possibility of adapting the diffuser to different values of the rotational speed.

For every given value of the velocity OC inside the rotor, a maximum value of the deviation δ exists for which the wave pattern previously considered is possible. When δ increases, the possibility of reflection of the wave EF at E (fig. 12(a)) becomes more critical, and for δ larger than a maximum value depending on the intensity of OC the reflection is no longer possible. The waves EF and ED form a lamda shock with a normal shock at E if the intensity of the compression waves from the stator continues to increase. The phenomena are similar to those found in supersonic diffusers. If the intensity of the compression waves increases, the strong shock moves upstream inside the rotor.

If the passage of the rotor is a diverging or a converging-diverging channel, the strong shock can be pushed upstream gradually by changing the geometry of the diffuser and increasing the intensity of the compression waves. If the passage does not have a contraction, the strong shock cannot be stabilized inside the passage; the disturbance moves in front of the rotor, changes the conditions in front of the rotor, and decreases the entrance axial velocity and, therefore, decreases the mass flow.

When the normal shock moves upstream in the rotor, the velocity OC' becomes subsonic and again has the direction OC. To every position of the shock inside the rotor there corresponds a given value of the velocity in front of the stator and, therefore, the corresponding stator configurations that push the shock near the minimum section can be determined. The axial velocity in front of the stator is subsonic, and the entrance velocity is parallel to the surface PQ.

In this analysis, several simplifying assumptions have been made. For example, the distance between rotor and stator is assumed to be sufficiently large to permit neutralization of the compression and expansion waves in front of the stator. The possibility of neutralization of waves inside the stator, however, could not greatly change the phenomena considered because the neutralization would occur automatically as a consequence of the relative motion between stator and rotor.

In the discussion it is assumed also that the flow inside the rotor is not changed when waves exist at the exit of the rotor passage. Actually, however, the presence of the boundary layer and of a zone of separation inside the passage permits the pressure disturbances to travel upstream in the rotor and to change to some extent the magnitude of the vector OC.

PRELIMINARY TESTS OF STATOR PASSAGE

Tests of two-dimensional variable-geometry diffusers which can be used in supersonic compressors of the type considered in the present paper are in progress at the Langley Laboratory. Preliminary results are presented in figures 13, 14, and 15 for Mach numbers of 2.51 and 2.01 for diffusers having supersonic and sonic axial entrance velocity components. In the inlets considered, no limitations of contraction ratio exist because variable geometry is used; therefore, if it is assumed that no interference exists between the boundary layer and the supersonic stream, configurations can be designed for which the losses due to supersonic compression are theoretically very small. With this assumption, diffuser configurations having gradual compression might be expected to yield higher pressure recovery. These and other experimental results (reference 3), however, indicate that the presence of the boundary layer affects the compression inside the diffuser to a large extent; therefore, in order to obtain a configuration that gives high pressure recovery, a compromise must be made between shock losses and losses due to boundary-layer separation.

For $M = 2.50$, in order to have high pressure recovery, compression ratios of the order of 12 or 15 are required in the stream. When the compression occurs gradually inside the inlet the boundary layer is subjected to large adverse pressure gradients and, therefore, it tends to separate. In order to avoid large pressure gradients it is more efficient to compress the flow to some extent with shocks in front of the inlet before the boundary layer is formed than to compress the flow gradually entirely inside the inlet. In either case, the amount of compression that can be obtained by means of variable geometry is determined by the boundary-layer phenomena. The test results lead to the conclusion that large compression at the stator entrance can be used efficiently only if the axial component of the velocity in front of the stator is supersonic. In this case, the compression can be obtained by producing a shock at the leading edge A of one of the blades and by reflecting the shock near the leading edge of the other blade B (figs. 13 and 14). The tests show that when the reflected shock hits the surface AC of the diffuser it tends to produce separation and, therefore, it is desirable to reduce the intensity of the reflected shock to small values. The minimum value of the intensity of the reflected wave is determined by the structural requirements of the blades because the wedge angle of the blade at the leading edge controls the intensity of the reflected shock. For example, if the shock from the leading edge A corresponds to a deviation of 14° and the reflected shock to a deviation of 9°, the blade wedge is 9°.

Tests have been made at $M_5 = 2.51$ for the diffuser shown in figure 13(a) at different angles of attack in order to change the intensity of the compression in front of the passage. The results of these tests show that when the shock in front of the inlet increases, the pressure recovery increases. The maximum pressure recovery has been obtained at the highest angle of attack of the model tested (fig. 13(a)), for which the shock from the leading edge A of the blade corresponds to a deviation of 14°. For this condition, the upper surface has a direction that gives a 9° wedge angle for the blade (fig. 13(b)). The blade tested has a 2-inch span and the boundary layer was removed from the side walls in perpendicular direction to the stream 1 inch in front of the blade AC. The maximum pressure recovery obtained for the diffuser configuration shown in figure 13(a) is of the order of 74 percent. Tests were made at the same Mach number with a diffuser having gradual curvature along AC, but satisfactory values of pressure recovery could not be obtained.

The inlet of figure 14(a) at $M_5 = 2.01$ had a pressure recovery of about 0.88. Figure 14(b) shows a possible blade shape for the stator that corresponds to this test model. When sonic or subsonic axial entrance velocity at the stator is considered, the blade must have the surface AC parallel in the zone near the leading edge to the stream direction if all disturbances are to be confined in the stator passage. Therefore, in order to have a practical wedge angle at the leading edge of the blade, a wave from the other surface must be produced of the same strength as in the case of the supersonic axial component. The compression in this case occurs in a zone between the walls in which boundary layer exists; whereas, in the supersonic case, the compression across the first shock occurs without adverse effects from the wall boundary layers. In the subsonic axial-component case, a diffuser of the type shown in figure 15 can be used. The diffuser in the zone ABCD is similar to the diffuser of figure 13 except for a surface AA', parallel to the incoming stream, that is added to the entrance. In this inlet, however, the inclination of the surface BD with respect to the incoming stream must be larger than in the inlet of figure 13 because in the inlet of figure 13 the wedge at the leading edge of the blade is fixed by the inclination between the surfaces AC and BD; whereas, in this inlet it is given by the inclination between the line BD and AA'. As the subsonic axial component of the velocity entering the stator is decreased the length AA' will obviously increase. In order to keep the leading-edge section practical structurally it will probably be essential to increase the wedge angle as the axial velocity component decreases.

Tests have been performed at $M = 2.51$ for the diffuser shown in figure 15(a) in which the surface AA' was omitted (the diffuser is the same as the diffuser of fig. 13(a), but has a smaller angle of attack such that the axial component is sonic, and the movable 5° flap is adjusted so that its upper surface is parallel to the entering flow). For the configuration of figure 15(a), the maximum value of pressure recovery obtained with variable geometry was of the order of 0.59. When the intensity of the shock from A was increased, higher pressure recovery was obtained; however, the wedge angle at the leading edge necessarily decreases as the shock from A increases in strength. For example, for the configuration of figure 13(a), the wedge would have negative values. The low values of pressure recovery for the sonic axial-component configuration (fig. 15(a)) can be attributed to the increased intensity of the shock from BD which increases the separation at the surface AC and, therefore, increases the losses and instability of the diffuser.

No tests have been performed for the lower Mach numbers with subsonic axial component which are required for the stator of the type-2 compressor. In this case, the entering Mach number M_5 is small (of the order of 1.50) and high pressure recovery can be expected.

The Reynolds number of these tests was considerably higher than usually found in compressors (of the order of 3.7×10^6 per inch) with the model blade spacings of the order of 1 inch. Because of the importance of the boundary layer as it affects diffuser performance, the test Reynolds number and aspect ratio are parameters which should be investigated in future tests.

These results are preliminary but seem to indicate that in the velocity diagram a supersonic axial-velocity component in front of the stator is preferable to subsonic components. Advantages can be expected by the removal of the boundary layer along the upper and lower surfaces of the annulus in front of the stator.

CONCLUDING REMARKS

A supersonic compressor design having supersonic velocities at the inlet to the stator was analyzed on the assumption of two-dimensional flow. Preliminary supersonic cascade tests of the rotor and stator blades are described from which values of the rotor and stator losses were determined for use in the analysis. The results of the calculations indicated that stage compression ratios between 6 and 10 can be obtained for this design with adiabatic efficiency between 70 and 80 percent.

The starting conditions and stability of the flow in rotor and stator are discussed, and the desirability of variable-geometry stators and adjustable guide vanes is indicated. These devices increased the ranges of entrance velocity and rotational speed for high efficiency. Although either supersonic or subsonic axial entrance velocity to the stator can be used in this design, the preliminary cascade tests indicated that supersonic velocity results in higher pressure recovery.

Langley Aeronautical Laboratory
National Advisory Committee for Aeronautics
Langley Air Force Base, Va.

REFERENCES

1. Kantrowitz, Arthur: The Supersonic Axial-Flow Compressor. NACA ACR L6D02, 1946.

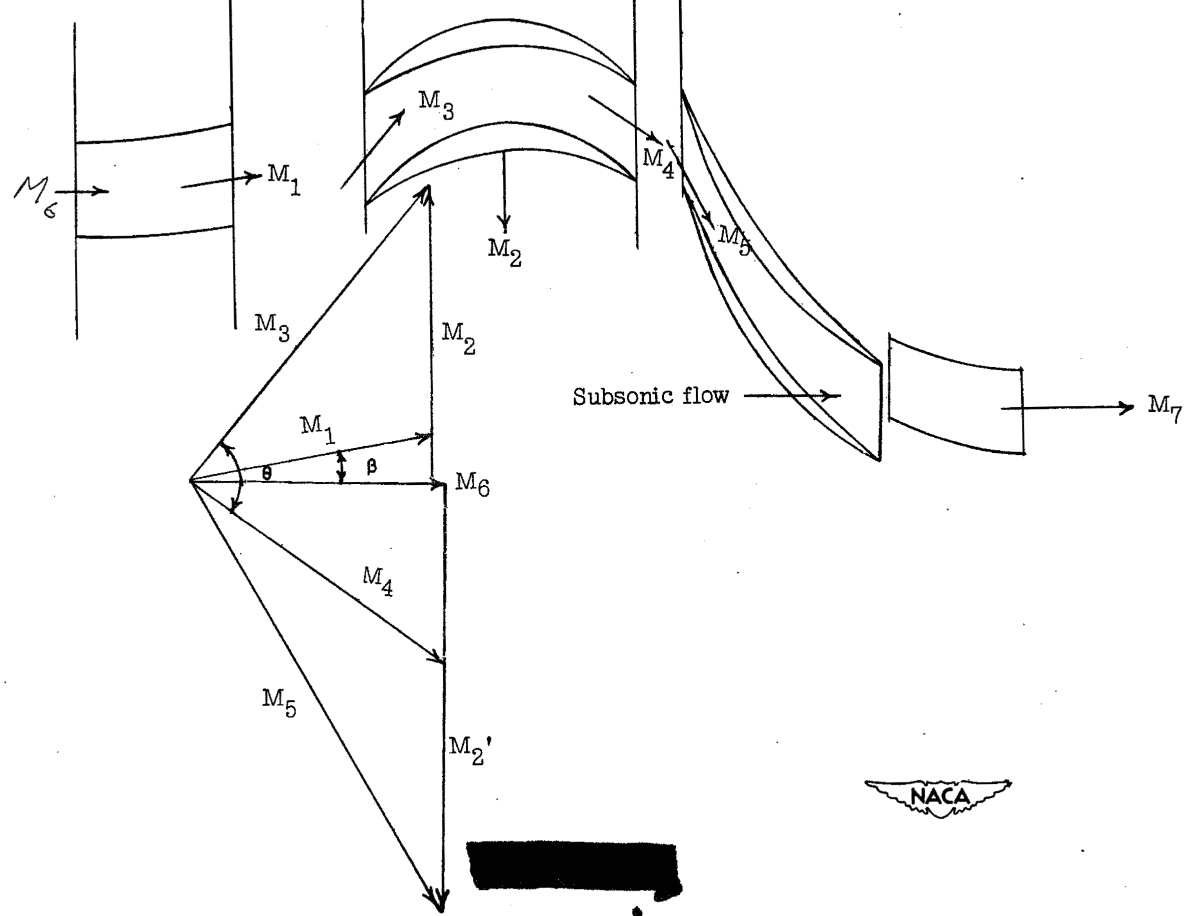
2. Wright, Linwood C., and Klapproth, John F.: Performance of Supersonic Axial-Flow Compressors Based on One-Dimensional Analysis. NACA RM E8L10, 1949.

3. Ferri, Antonio, and Nucci, Louis M.: Theoretical and Experimental Analysis of Low Drag Supersonic Inlets Having a Circular Cross Section and a Central Body at Mach Numbers of 3.30, 2.75, and 2.45. NACA RM L8H13, 1948.

4. Liccini, Luke L.: Analytical and Experimental Investigation of 90° Supersonic Turning Passages Suitable for Supersonic Compressors or Turbines. NACA RM L9G07, 1949.

5. Ferri, Antonio: Experimental Results with Airfoils Tested in the High-Speed Tunnel at Guidonia. NACA TM 946, 1940.

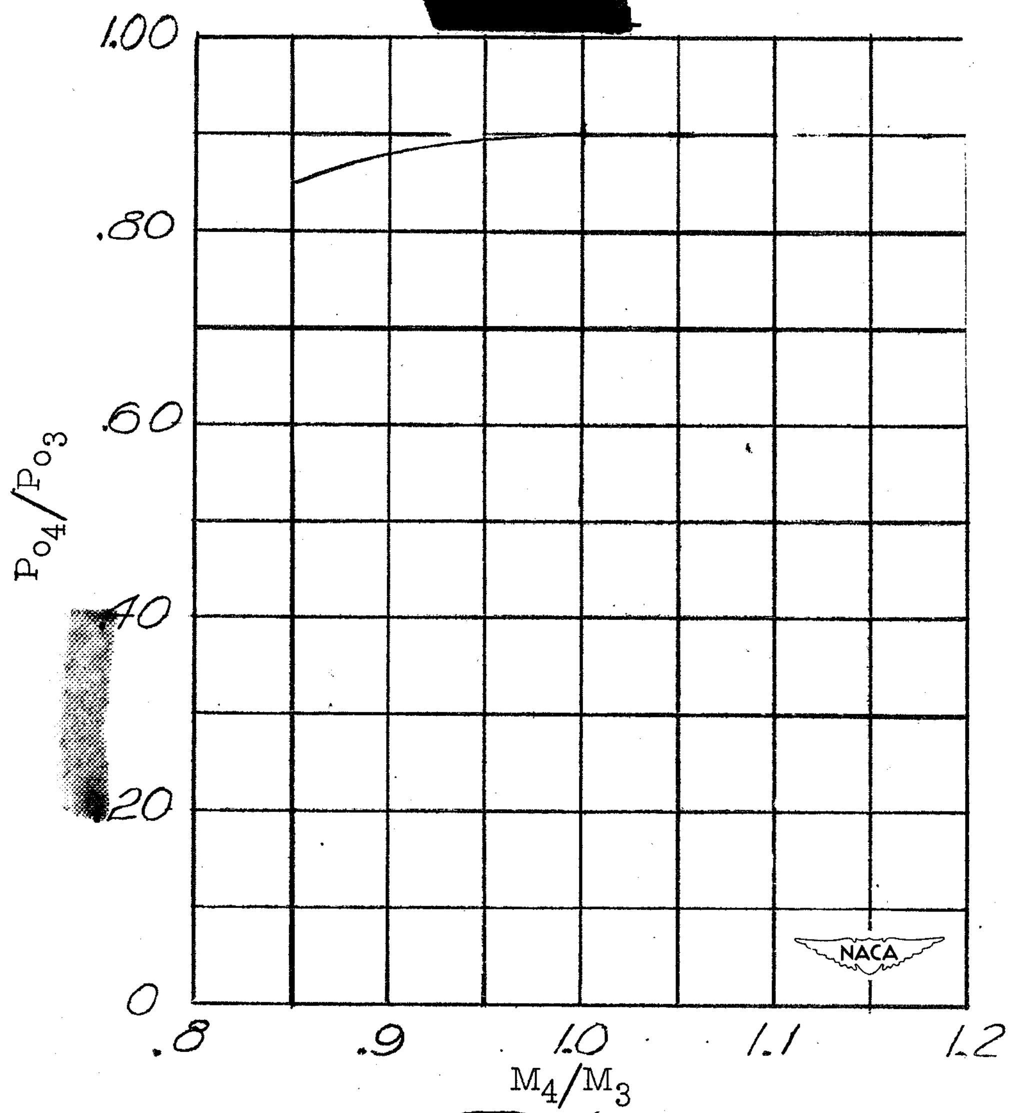
[redacted]



[redacted]

Figure 1.- Velocity diagram for a supersonic compressor having supersonic flow at the entrance of the stator.

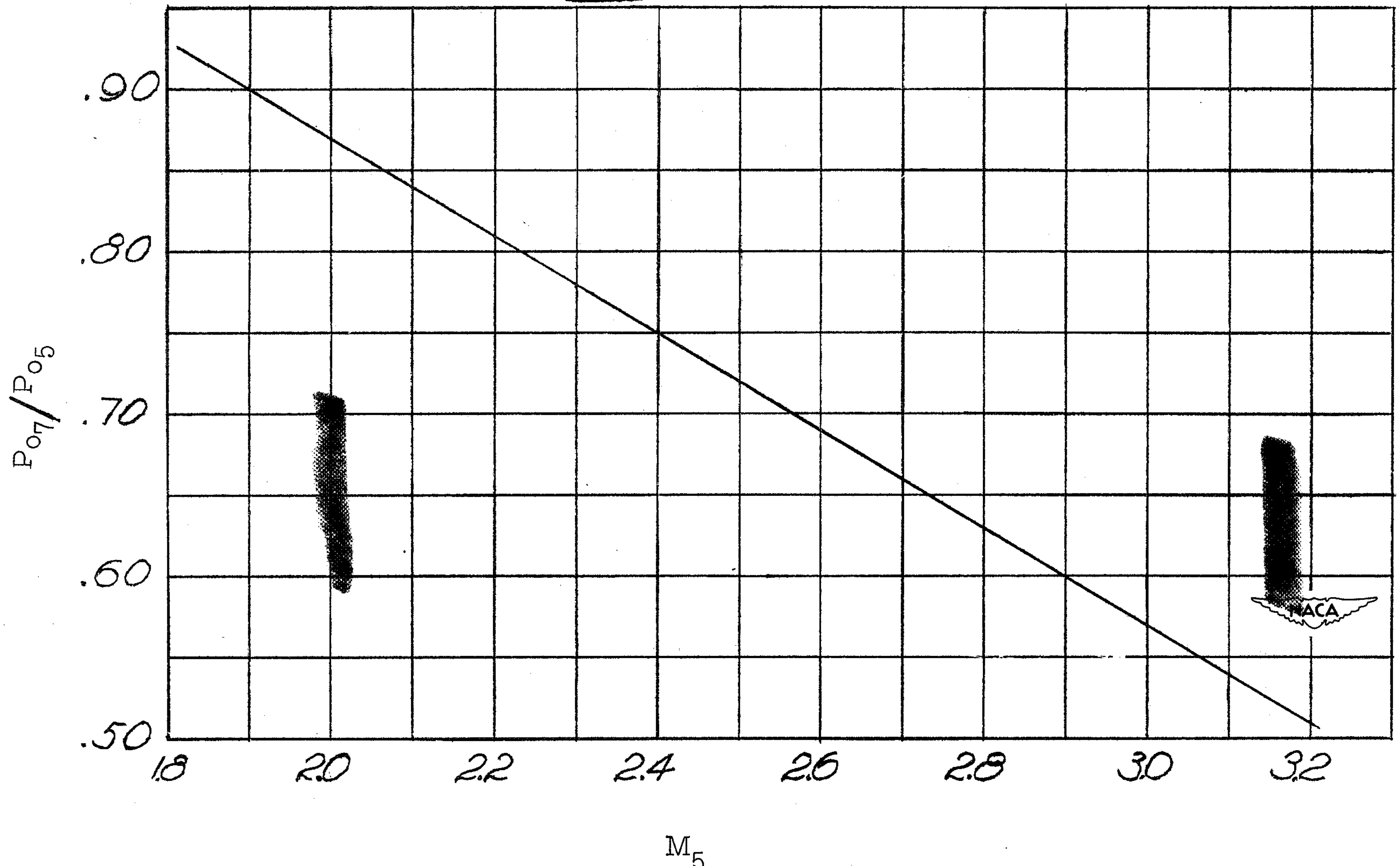
[redacted]



[redacted]

Figure 2.- Total pressure recovery assumed for the turning passage as a function of the ratio of the relative Mach number at the exit and at the entrance of the passage.

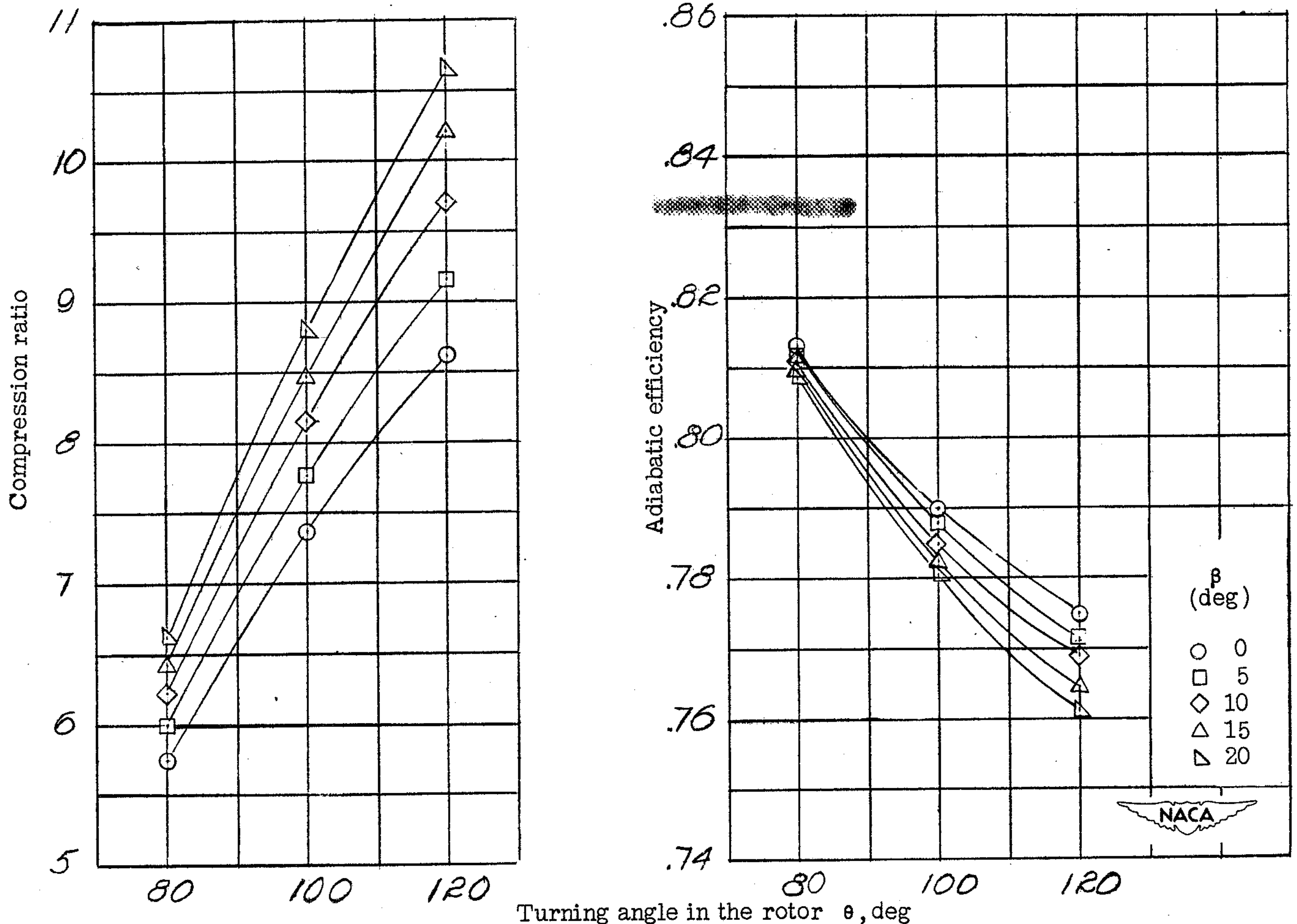
[redacted]



[redacted]

Figure 3.- Total pressure recovery assumed for the stator as a function of the entrance Mach number M_5.

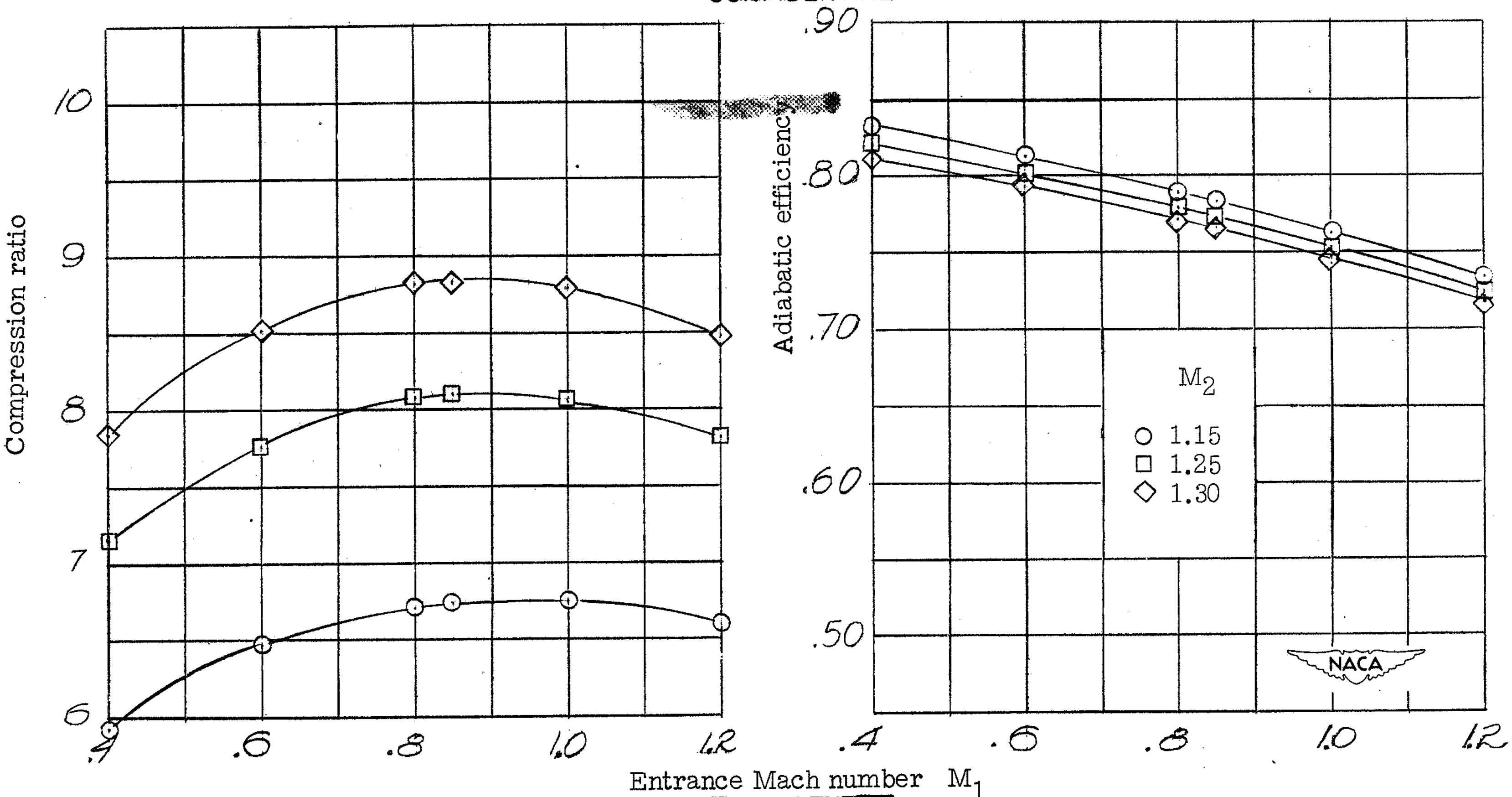
[redacted]



[redacted]

Figure 4.- Variation of the compression ratio and adiabatic efficiency with the turning angle in the rotor for different values of turning in the entrance vanes. Rotational Mach number, $M_2 = 1.25$; entrance Mach number, $M_1 = 0.85$; compression in the rotor, $\frac{M_4}{M_3} = 0.90$.

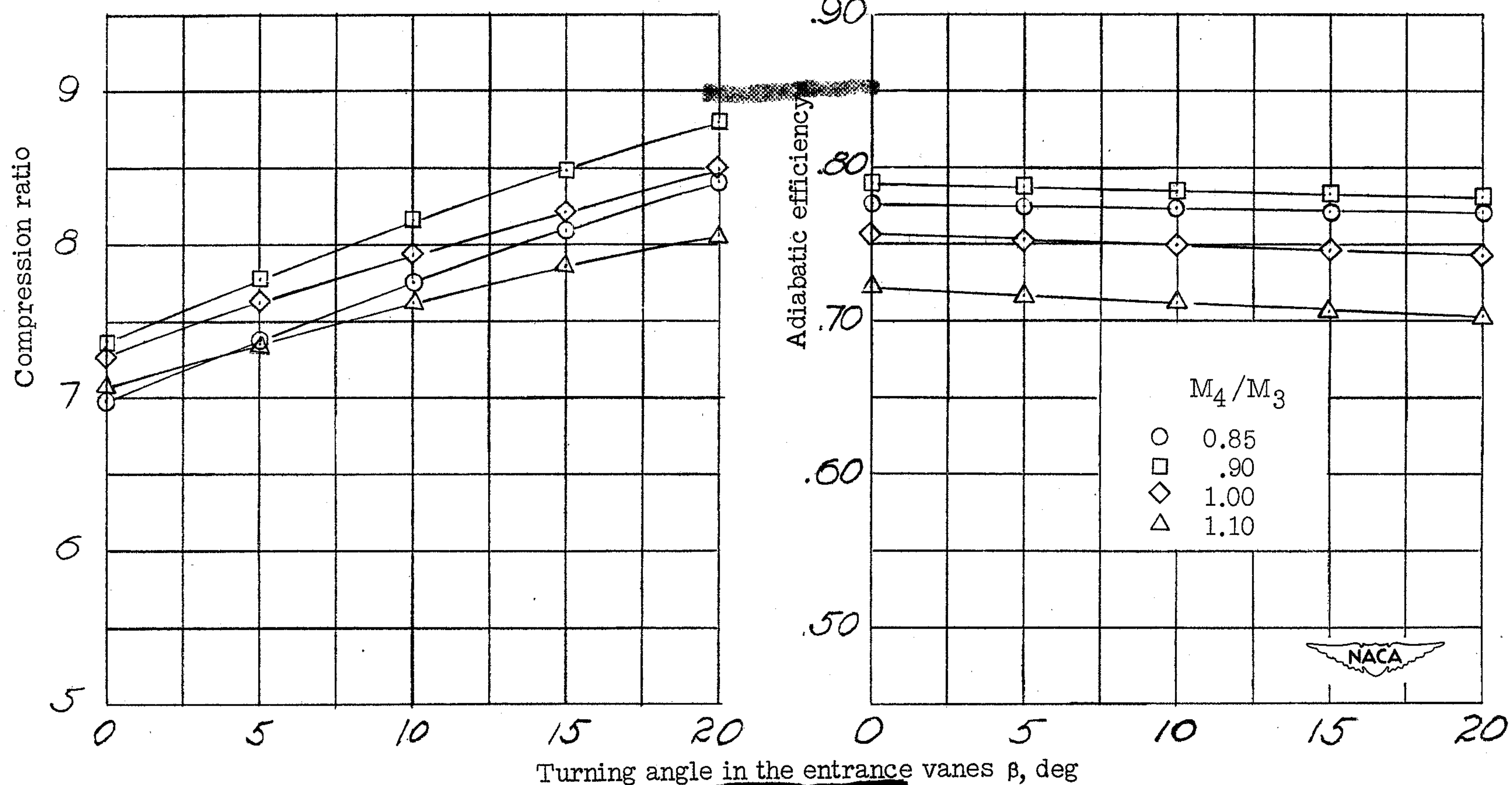
[redacted]



[redacted]

Figure 5.- Variation of the compression ratio and adiabatic efficiency with the entrance Mach number M_1 for different values of rotational Mach number M_2. Turning angle in the entrance vanes, $\beta = 15^{\circ}$; turning angle in the rotor, $\theta = 100^{\circ}$; compression in the rotor, $\frac{M_4}{M_3} = 0.85$.

[redacted]

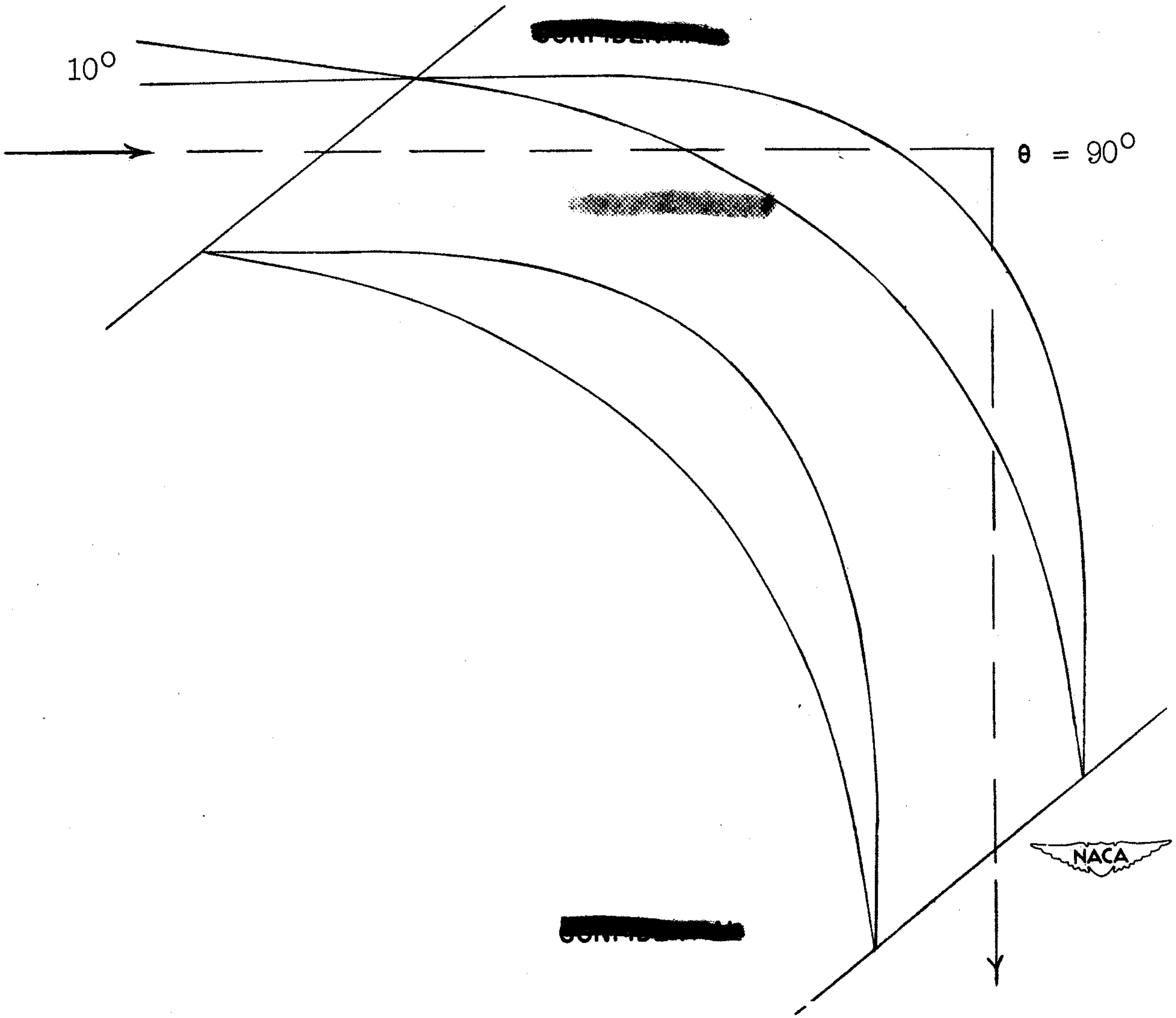


[redacted]

Figure 6.- Variation of compression ratio and adiabatic efficiency with the turning angle in the entrance vanes β for different values of compression in the rotor. Turning angle in the rotor, $\theta = 100^{\circ}$; rotational Mach number, $M_2 = 1.25$; entrance Mach number, $M_1 = 0.85$.

[redacted]

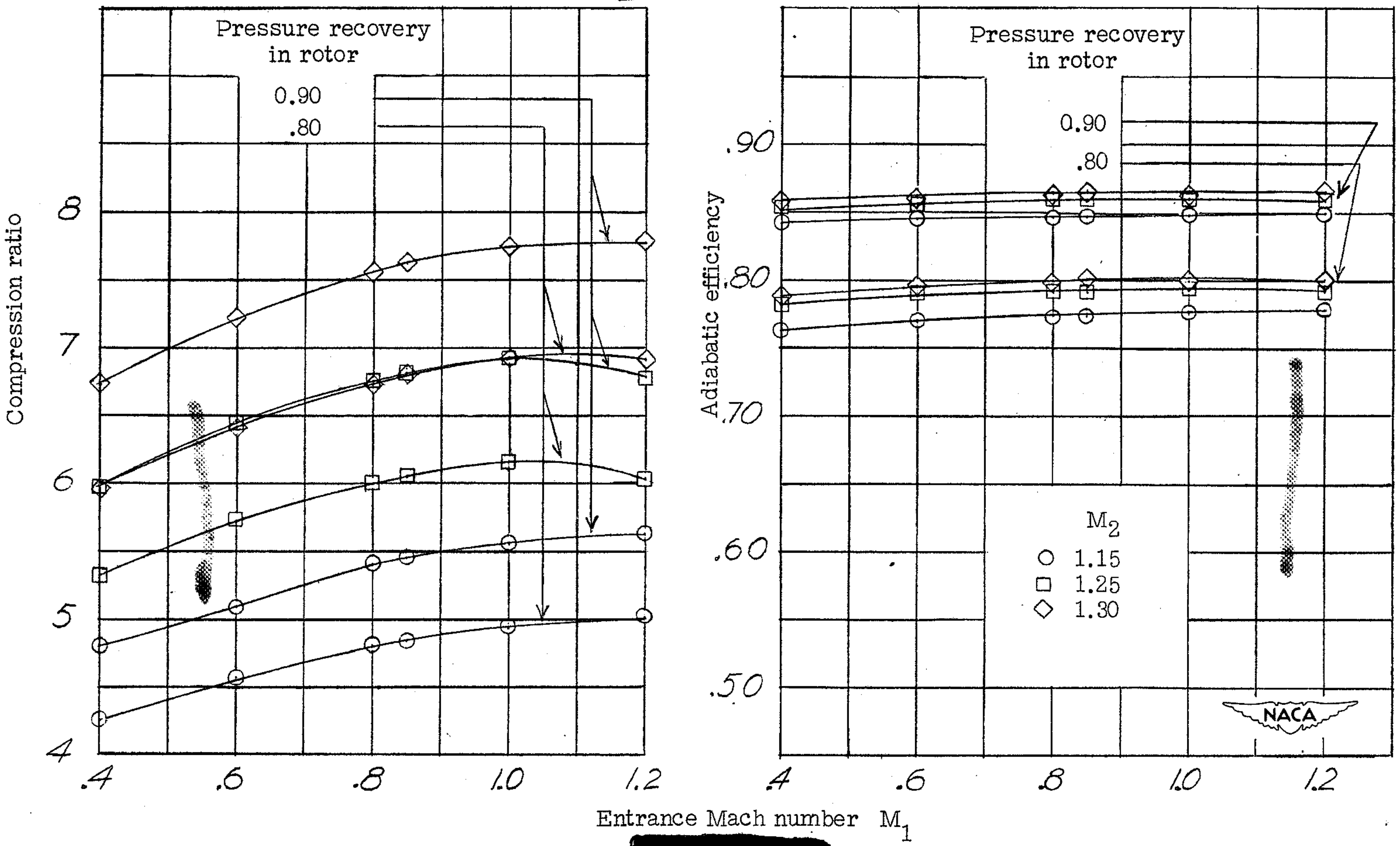
[redacted]



[redacted]

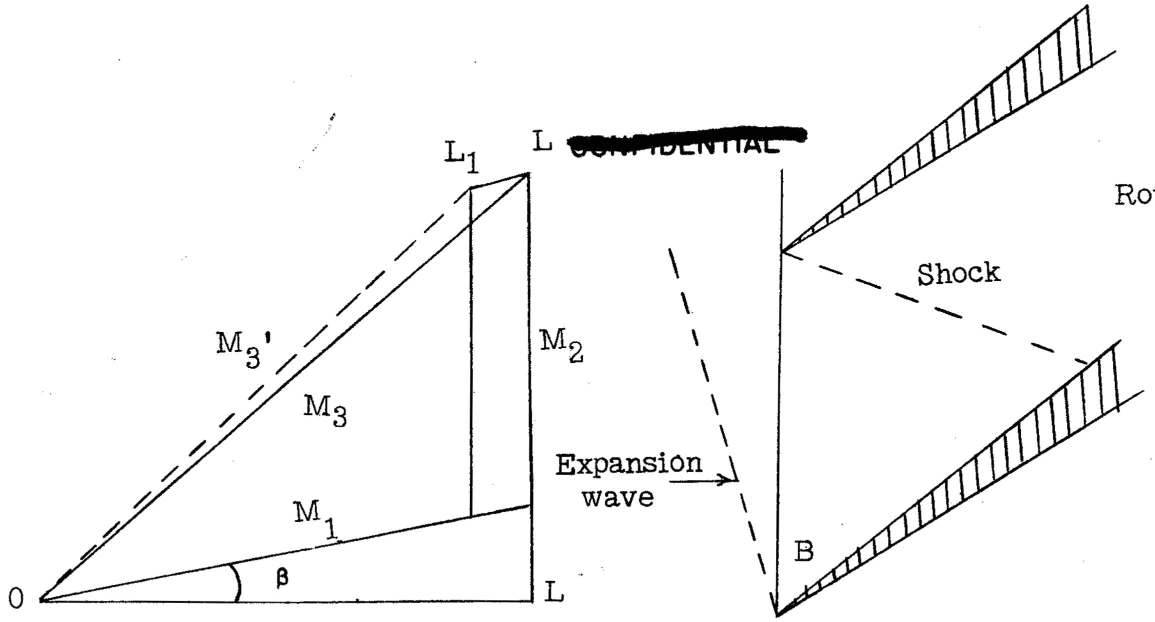
Figure 7.- Rotor blade design for $M_3 = 1.71$ (model 3 of reference 4).

[redacted]

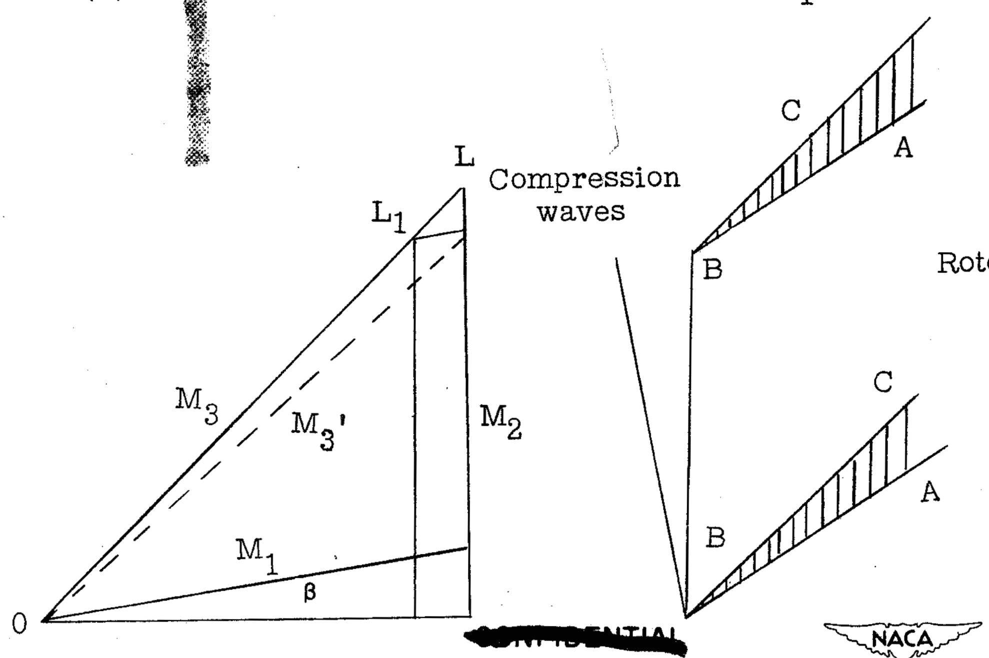


[redacted]

Figure 8.- Variation of compression ratio and adiabatic efficiency with the entrance Mach number M_1 for different values of rotational Mach number M_2, and pressure recovery in the rotor. Turning angle in the entrance vanes, $\beta = 15^{\circ}$; turning angle in the rotor, $\theta = 100^{\circ}$; compression in the rotor, $\frac{M_4}{M_3} = 0.37$.



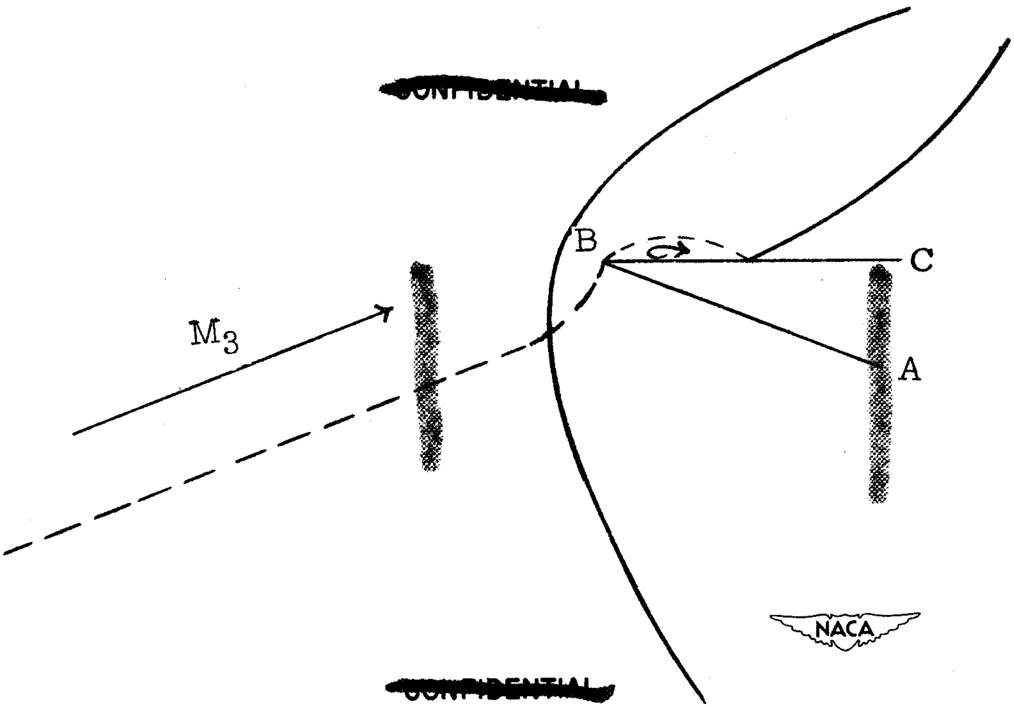
(a) Variations of the entrance Mach number M_1.



(b) Variation of rotational velocity.

Figure 9.- Stability of the rotor ($M_1 \cos \beta < 1$).

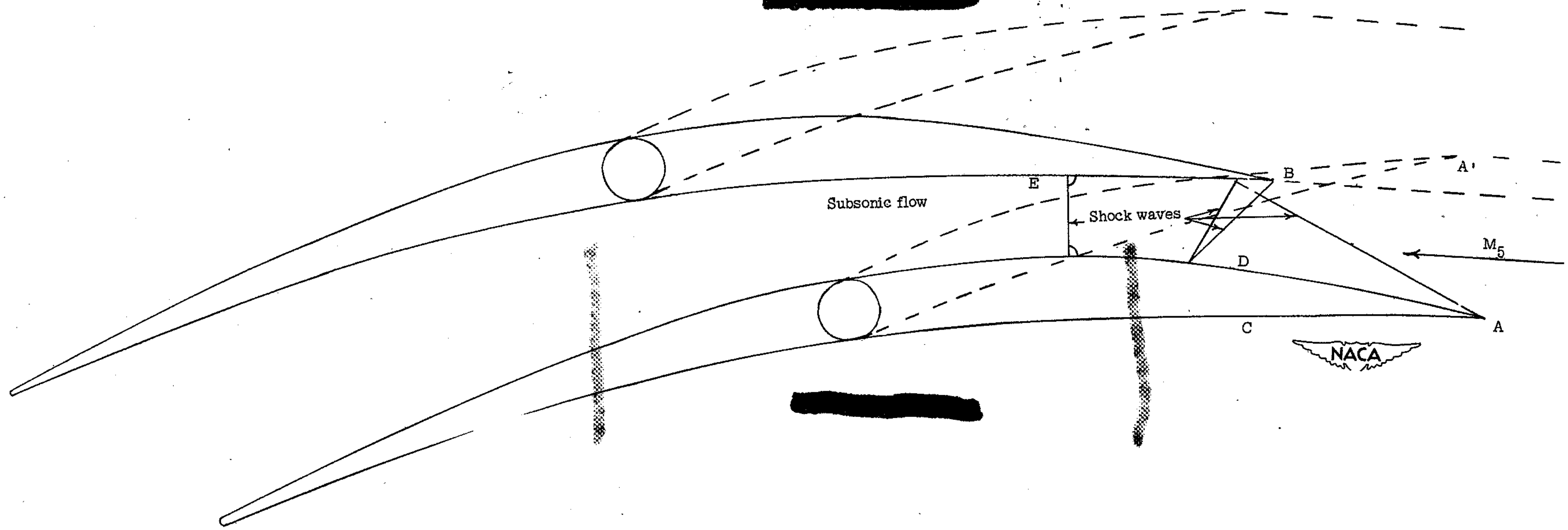
[redacted]



[redacted]

Figure 10.- Detached shock at the leading edge of the rotor blade.

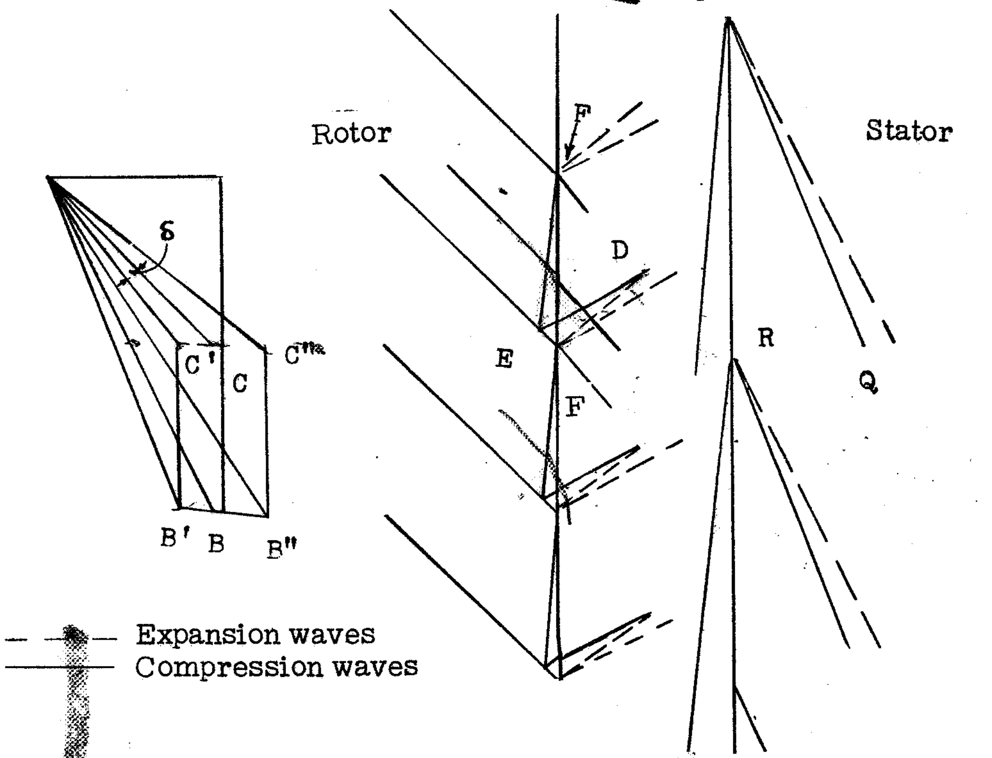
[redacted]



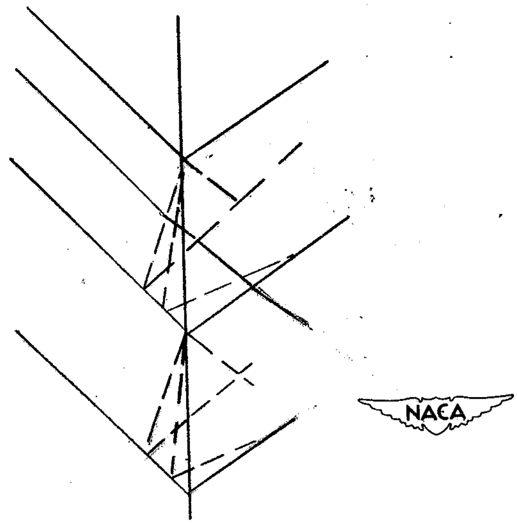
[redacted]

Figure 11.- Variable-geometry stator blade.

[redacted]



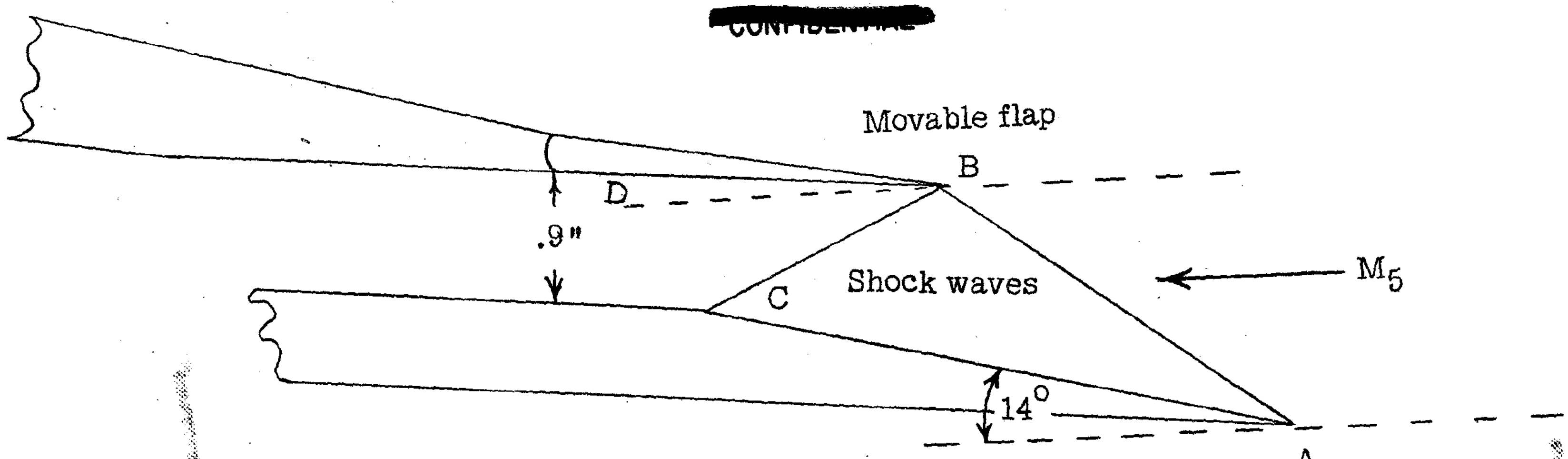
(a) Compression waves moving upstream from the stator.



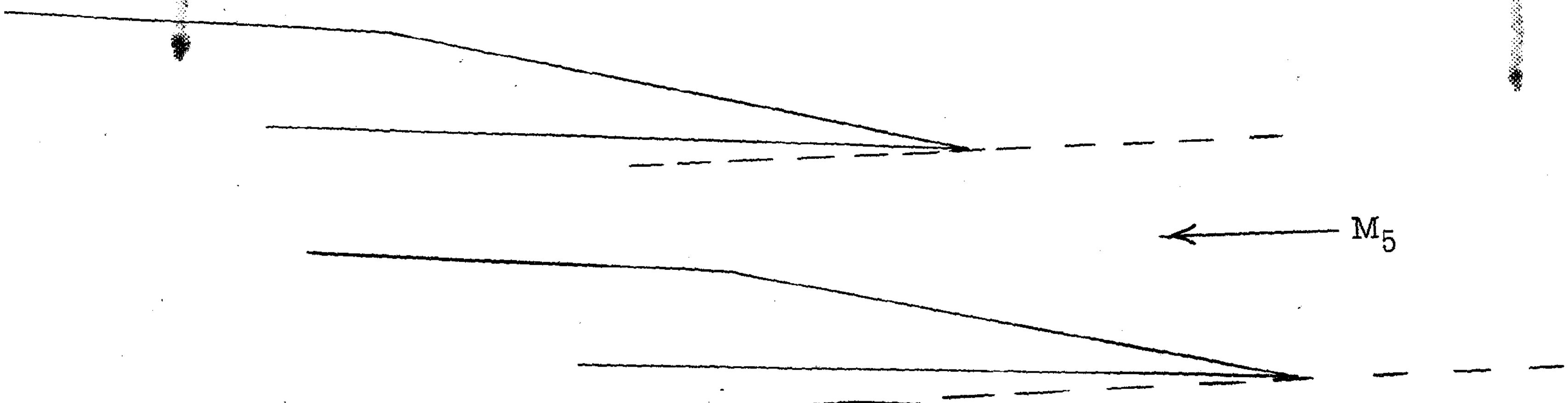
[redacted]

(b) Expansion waves moving upstream from the stator.

Figure 12.- Interference between rotor and stator.

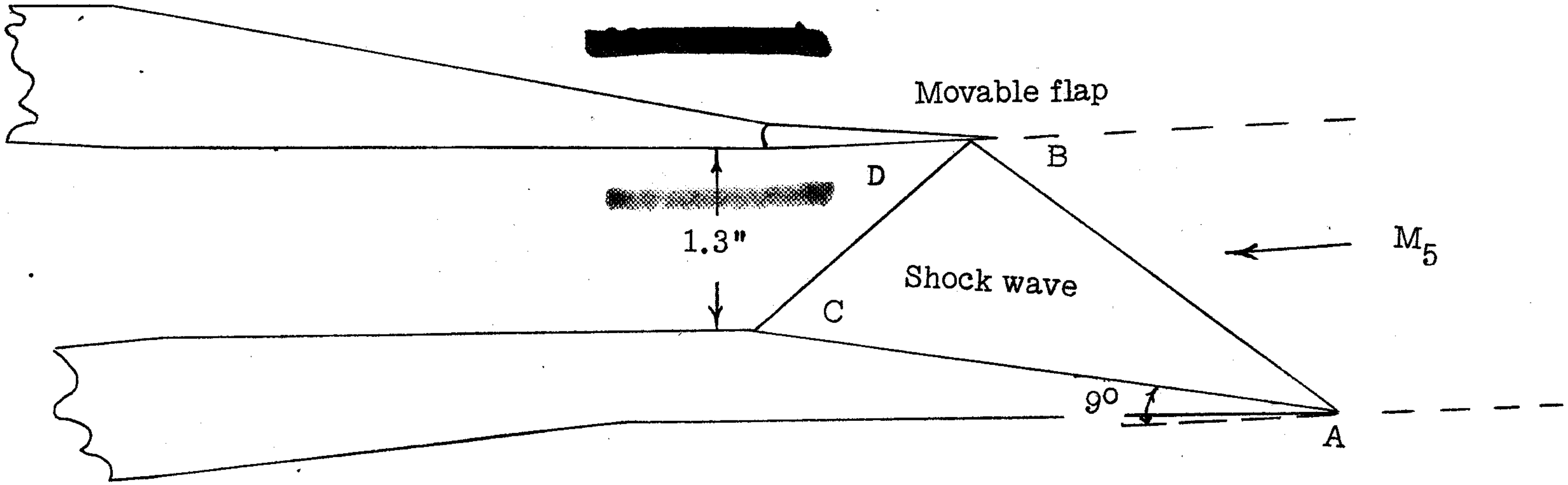


(a) Model of stator blade tested at $M_5 = 2.51$.

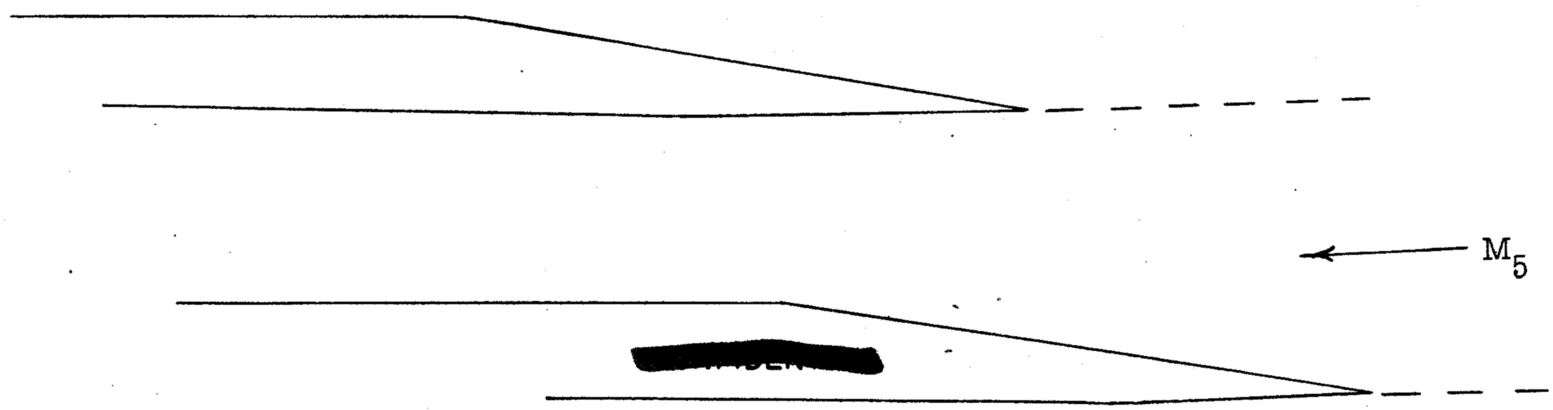


(b) Stator passage corresponding to model of figure 13(a).

Figure 13.- Stator for $M_5 = 2.51$.



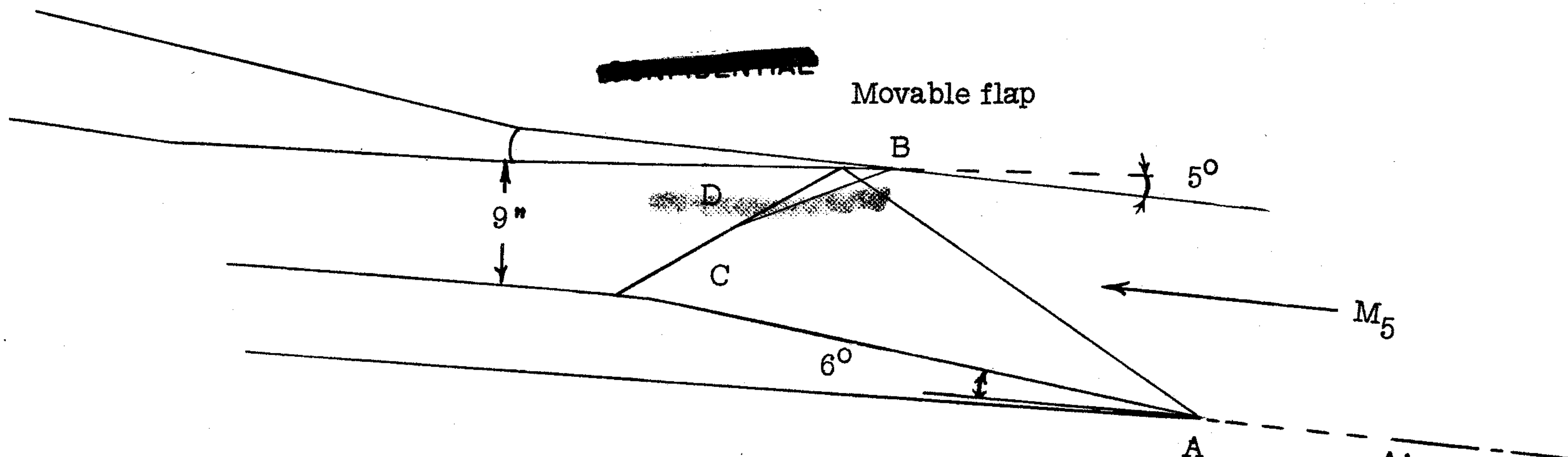
(a) Model of stator blade tested at $M_5 = 2.01$.



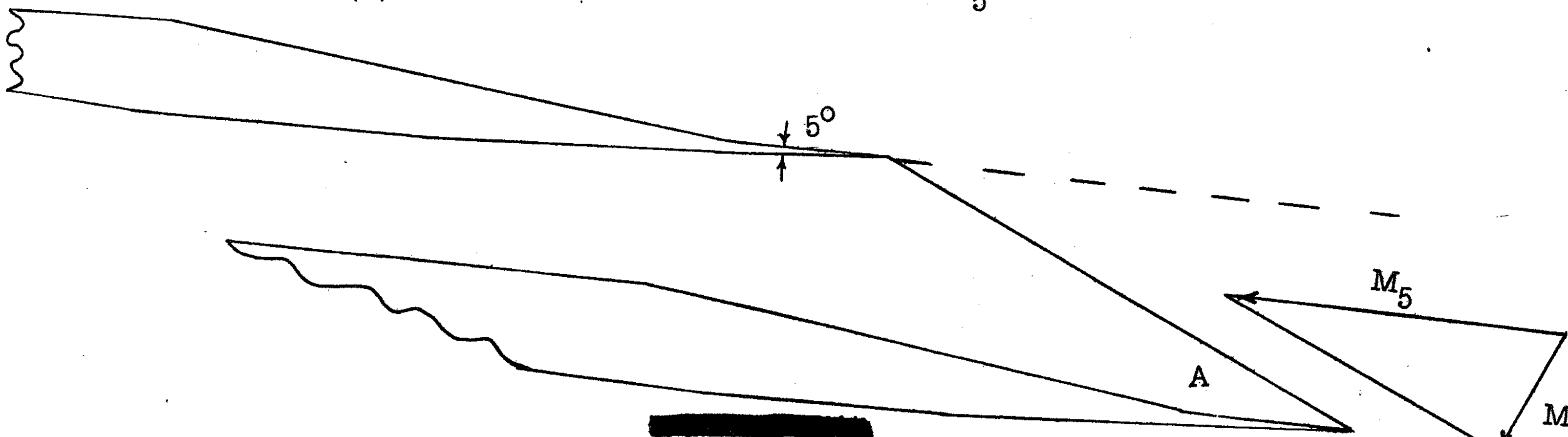
(b) Stator passage corresponding to model of figure 14(a).

Figure 14.- Stator for $M_5 = 2.01$.

[redacted]



(a) Model of the stator blade tested at $M_5 = 2.51$.



[redacted]

(b) The stator passage corresponding to the model of figure 15(a). $M_5 = 2.51$ ($M_6 = 1$).

Figure 15.- Stator for $M_5 = 2.51$ ($M_6 = 1$).